APOLLO NAVIGATION WORKING GROUP

TECHNICAL REPORT

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APOLLO MISSIONS AND NAVIGATION SYSTEMS CHARACTERISTICS

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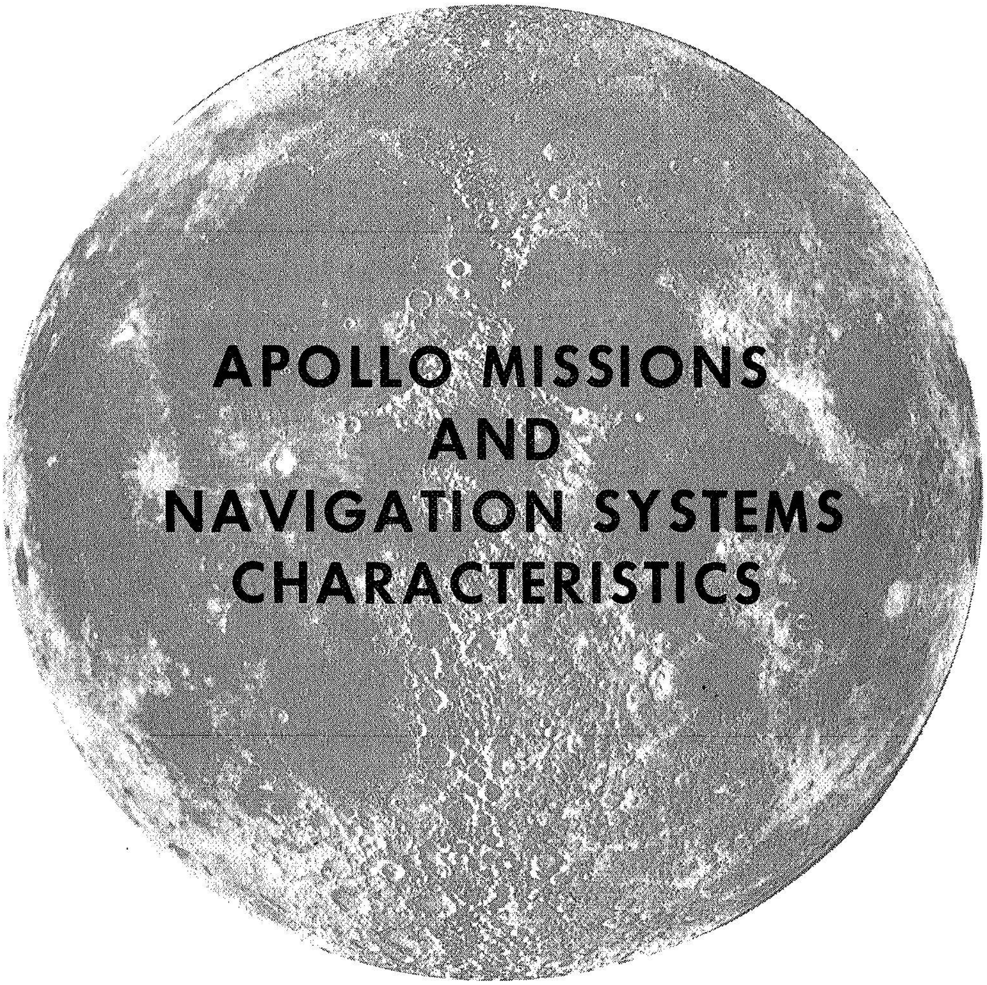
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

APOLLO NAVIGATION WORKING GROUP

TECHNICAL REPORT



APOLLO MISSIONS AND NAVIGATION SYSTEMS CHARACTERISTICS

F. O. Vonbun
Chief, Systems Analysis Office
Goddard Space Flight Center
Greenbelt, Maryland

John P. Mayer
Chief, Mission Planning and Analysis Division
Manned Spacecraft Center
Houston, Texas

Contributors

Goddard Space Flight Center	Manned Spacecraft Center
D. Anderson	A. Bond
J. Barsky	V. Bond
P. Brumberg	S. Crossland
J. Cooley	J. Dalby
P. Emmons	J. McPherson
R. Groves	P. Pixley
C. Jacobs	R. Savely
T. Jones	E. Schiesser
W. Kahn	S. Shaper
B. Kruger	J. Taylor
A. Marlow	W. York
R. Sandifer	

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1.0 INTRODUCTION

The purpose of the Apollo Navigation Working Group is to coordinate, analyze and study problems of the Apollo missions from the navigation point of view. The problem areas include the onboard navigation and communication systems as well as the world-wide ground tracking network. Emphasis is placed on analysis of the total "system" rather than the onboard and ground systems since the combination of both guarantees the success of the mission. Based upon these analyses, it is the group's responsibility to recommend changes in both the onboard and ground systems in order to fulfill the mission.

The information is divided into four areas: The Apollo missions, the trajectory prediction capability, the measurement and communication systems, and the data processing procedures. Development in all of these areas is not complete. Thus, the information represents the best current knowledge and will be updated as required.

For consistency with the usage in the Apollo program, international English units will be used in this document with the exception that certain constants in Chapter 4 may be given in units consistent with their definitions or common usage. Metric units will be included in parentheses. However, for expediency in preparing this first issue, some tables throughout the report will be given in units other than the international English ones. The preferred international English units are:

length	nautical mile (nm) or foot (ft)
speed	foot per second (ft/s)
mass	pound (lb or lbm)
force	pound (lb or lbf)

The ANWG will compile three documents. The first, the "Apollo Missions and Navigation Systems Characteristics" will contain information for use in the study of Apollo navigation capability. The second, "Apollo Navigation – Ground and Onboard Capabilities," will summarize results of studies of navigation capability. In both of these documents an attempt will be made to indicate the quality of the material presented. The third document will be a compilation if the individual reports from which the first two documents were compiled. The title will be: "Appendix, ANWG – Compilation of Source Documents."

2.0 CHANGES AND ADDITIONS

This document is the first publication of the Apollo Missions and Navigation Systems Characteristics Document. In future issues, this section will consist of a summary of the additions and revisions to the document.

3.0 APOLLO MISSIONS

The purpose of this chapter is to indicate the geometrical properties of the Apollo early missions and the Apollo lunar mission. Two specific reference trajectories of the lunar orbital mission are included.

3.1 Earth Orbital Missions

Earth orbital missions are the early Apollo missions whose objectives are the qualification of spacecraft systems and the practice of lunar orbital rendezvous techniques.

The following description of the objectives, configuration, and flight plan for the early missions gives an idea of recent thinking of types of missions being considered for these development flights. Although every effort is being made to keep the content of this document current, the ANWG can not be responsible for stating the most current concepts for these early missions on a continuing basis, as the concepts change more rapidly than the revision schedule for this document. The Apollo Spacecraft Program Office should be consulted for the up to date early mission details.

3.1.1 Mission 201[1]

Unmanned lob shot down ETR[2]
(First Saturn IB/Apollo Flight)
Hi heat rate lifting entry; $V_i \approx 28{,}000$ fps
CM[3] splash close to Ascension Island

3.1.2 Mission 202

Unmanned, long-range lob shot
Hi heat load lifting entry ($V_i \approx 28{,}000$ fps)
CM splash near Wake Island

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[1] 200 Series denotes Saturn IB launch vehicle; 500 Series denotes SV launch vehicle.

[2] ETR – Eastern Test Range

[3] CM – Command Module

3.1.3 Mission 203

Deleted

3.1.4 Mission 204

First manned CSM[4] flight
Earth orbital, open-ended up to 14 days
Shallow lifting entry, $V_i \approx 25{,}000$ fps

3.1.5 Mission 205

Manned CSM long duration mission
Earth orbital
(Mission profile not yet established)

3.1.6 Mission 206

LEM[5] development (unmanned launch)
Complete, full LEM; low earth orbit
12 hours maximum duration; no recovery

3.1.7 Mission 207

First manned CSM-LEM flight
Earth orbital, open-ended after 3 days
Manned LEM separation and operations
CSM Block II[6] subsystems verification

3.1.8 Mission 501

CM lunar return reentry verification
S/C[7] inserted into 12,000 n.mi. ellipse by Saturn V
SPS[8] burns provided CM reentry conditions
12 hour total duration, 2 parking orbits

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[4]CSM – Command and Service Module

[5]LEM – Lunar Excursion Module

[6]CSM Block II – Spacecraft configuration for missions 207, 503, 504, and any lunar mission; contrasted to Block. I, which is the designation of the spacecraft configuration to be used in the early missions.

[7]S/C – Spacecraft

3.1.9 Mission 502

CM lunar reentry verification
S/C inserted into 12,000 n.mi ellipse by Saturn V
SPS burns provided CM reentry conditions
12 hour total duration, 2 parking orbits

3.1.10 Mission 503

First manned Saturn V
Lunar mission simulation in earth orbit
10 days duration, 450 n.mi. apogee

3.1.11 Mission 504

First lunar landing mission

3.2 Lunar Mission

The lunar mission may be called that mission during which the vehicle enters the lunar sphere of influence. It is commonly divided into 13 phases. Within each phase the possible trajectories are limited by operational constraints, but they still offer an extensive selection. Factors governing the selection of a particular trajectory are the day of launch, delays, contingencies, and performance.

The following are guidelines to be used in constructing Apollo type lunar missions for navigational study. They apply to conic type or patched conic type construction. They list, by phase, the limiting constraints on the geometry of the trajectory. For the actual ground rules covering the operational mission plan construction, see MPAD document "Apollo Operational Nominal Trajectory Ground Rules."

These limiting constraints were determined by Mission Planning and Analysis Division. It should be noted that these guidelines are consistent with a ground rule requiring the selection of a free return translunar trajectory. The possibility of using a nonfree return translunar trajectory should not be ruled out. In this case, the translunar flight time could be as much as 110 hours.

[8]SPS – Service Propulsion System

The lunar mission will be divided into the following 13 phases:

- Launch
- Earth Parking Orbit
- Translunar Injection Burn
- Translunar Trajectory
- Lunar Hyperbolic Fly-by Trajectory
- Lunar Fly-by Transearth Trajectory
- Lunar Parking Orbit
- LEM Descent
- LEM Stay Time
- LEM Ascent
- Transearth Injection
- Transearth Trajectory
- Reentry and Recovery

3.2.1 Launch

Plane of the trajectory:

Launch pad coordinates:

Geodetic latitude = +28° 28' 56.92

Geodetic longitude = -80° 38' 08.07

The launch azimuth varies between 72° and 108°.

Insertion occurs 25° ±2° downrange in 720 seconds ±5 seconds after launch initiation.

3.2.2 Earth Parking Orbit

Plane of the trajectory: The plane of trajectory and the location of the insertion vector is defined by the launch.

Shape of trajectory: Typical trajectories are 100 n.mi. (185 km) circular. However, elliptical trajectories are not ruled out, in which case, perigee can be as low as 85 n.mi. (155 km) and apogee as high as 150 n.mi. (280 km).

Duration of the trajectory: One-half hour to four and one-half hours.

3.2.3 Translunar Injection Burn

The translunar injection will carry the vehicle down range from 21° to 25° in 305 to 325 seconds and will add approximately 60 n.mi. (110 km) in altitude.

The burn-out flight path angle varies between +1° and +7° and the burn-out velocity will vary between 35,750 fps (10,900 m/sec) and 35,450 fps (10,810 m/sec).

A plane change is limited to ±1°.

Injection is usually in the descending portion of the earth parking orbit if the moon is ascending and the ascending portion of the orbit if the moon is descending.

The burn-out conditions are determined by the particular translunar trajectory chosen.

3.2.4 Translunar Trajectory

Plane of the trajectory: Essentially the earth parking orbit plane.

Shape of the trajectory:

The intersection of the translunar trajectory plane with the earth-moon plane must coincide with the earth-moon line computed at the time of pericynthion arrival.

The location of the earth vacuum perigee vector will be ahead (ahead defined as in the direction of vehicle motion) of the computed earth-moon line by 6° to 11°. The shorter the flight time the further ahead.

The flight time from perigee to pericynthion arrival will lie between 60 hours and 75 hours, depending upon the earth-moon distance.

The translunar trajectory should allow a "free" return to the earth.

3.2.5 Lunar Hyperbolic Fly-by Trajectory

Plane of the trajectory:

The inclination of the trajectory plane with the earth-moon plane will be less than 15° for free return flight plans.

The vehicle must enter in a retrograde direction with respect to the moon's rotation upon its axis.

Shape of the trajectory:

Pericynthion altitude is between 75 n.mi. (140 km) and 85 n.mi. (155 km).

The intersection of the pericynthion vector with the lunar surface is between 180° and 195° longitude and ±20° latitude in the earth-moon plane coordinate system.

The velocity at pericynthion can vary between 7500 ft/sec (2290 m/sec) and 9000 ft/sec (2740 m/sec).

The lunar hyperbolic fly-by trajectory should allow a free return to the earth.

3.2.6 Lunar Fly-by Transearth Trajectory

Plane of the trajectory: There is no limitation on the inclination of the free-return transearth trajectory to the earth equatorial plane, as long as the vehicle makes a posigrade return; i.e., in the direction of the earth's rotational motion.

Shape of the trajectory:

The vacuum perigee altitude must be between 6 n.mi. (11 km) and 36 n.mi. (67 km).

The angle that the vacuum earth perigee vector makes from the earth-moon line, computed again at the time of pericynthion, will be between -6° and -11°. Positive degrees are counted in the direction of vehicle motion.

The flight time from pericynthion to entry will vary from between 60 hours and 75 hours, depending upon the earth-moon distance.

3.2.7 Lunar Parking Orbit

Plane of the trajectory:

The lunar parking orbit must contain the pericynthion vector of the lunar hyperbolic fly-by trajectory.

It must also contain a landing site vector on the earth side of the moon which has a longitude of between 315° and 45° and a latitude of between ±5° and -5° in selenographic coordinates.

The plane must also be oriented so that the lunar landing site does not move out of the lunar orbital plane more than 0.5° during the period of 3 hours to 39 hours after lunar orbit insertion.

Shape of the trajectory: The trajectory will be near-circular, with the semi-major axis being 1018.5 n.mi. (1886 km) and the insertion vector being along the lunar hyperbolic pericynthion vector, whose altitude is 80 n.mi. ±5 n.mi. (148 ±9 km).

Duration of the trajectory: Two hours to 60 hours.

3.2.8 LEM Descent

Plane of the trajectory: Coplanar with the lunar parking orbit.

Shape of the trajectory:

The LEM descent will be a Hohmann transfer from the lunar parking orbit to a perilune of 50,000 feet (8.23 n.mi.; 15.2 km).

The chosen landing site should occur between 175 n.mi. (325 km) and 220 n.mi. (405 km) down range from perilune position (assumed radius of 939.25 n.mi., 1739.5 km).

3.2.9 LEM Stay Time

Stay time on the lunar surface will be between 0 hours and 24 hours.

3.2.10 LEM Ascent

The perilune of the LEM ascent trajectory will be approximately 35,000 feet (5.76 n.mi., 10.7 km).

The trajectory will be an approximated Hohmann transfer with a transfer angle varying from 160° to 220°.

The inclination angle of the LEM ascent plane with the lunar parking orbit plane will generally be less than 0.5°.

3.2.11 Transearth Injection

The vehicle will not be visible to the earth for at least 20 minutes after termination of the transearth injection burn. For purposes of error analysis, it may be assumed that the vehicle is accelerated instantaneously from the proper lunar parking orbit to the proper outgoing transearth orbit.

3.2.12 Transearth Trajectory

Plane of the trajectory:

The inclination of the transearth trajectory plane with the earth equatorial plane will be between 0° and 40°.

The vehicle must make a posigrade return with respect to the earth's rotation.

The transearth injection from lunar parking orbit allows for a plane change.

Shape of the trajectory:

The vacuum perigee altitude and its associated uncertainties must lie between 6 n.mi. (11 km) and 36 n.mi. (67 km) with 25 n.mi. (46 km) being a nominal value.

The vacuum perigee vector will lie between -6° and -11° from the earth-moon line computed at the time of pericynthion, with positive degrees being measured in the direction of the vehicle's motion.

The transit time from injection to reentry will lie between 85 hours and 110 hours.

3.2.13 Reentry

Reentry will begin at 400,000 feet altitude (65.83 n.mi., 122 km).

The down range distance from reentry point to impact point will be planned to lie between 1500 n.mi. (2780 km) and 2500 n.mi. (4630 km) in the primary mode. In the back-up mode, dispersions of from 1400 n.mi. (2540 km) to 3500 n.mi. (6840 km) could be encountered.

Two landing areas are designated. See Figure 3-1. They take the shape of rectangles on the east side of the following two points.

Pearl Harbor	Pago Pago
Latitude 21.37° N	Latitude 14.25° S
Longitude 157.97° W	Longitude 170.75° W

The rectangles have a longitudinal width of 400 n.mi. (740 km) and a latitudinal length of 2400 n.mi. (4440 km) which is 1200 n.mi. (200 km) North and 1200 n.mi. (2200 km) South of the landing site. Portions of these rectangular landing areas which fall above or below ±40° latitude, respectively, are not to be used, as well as any areas in the near vicinity of land masses which occur in the general rectangular landing areas.

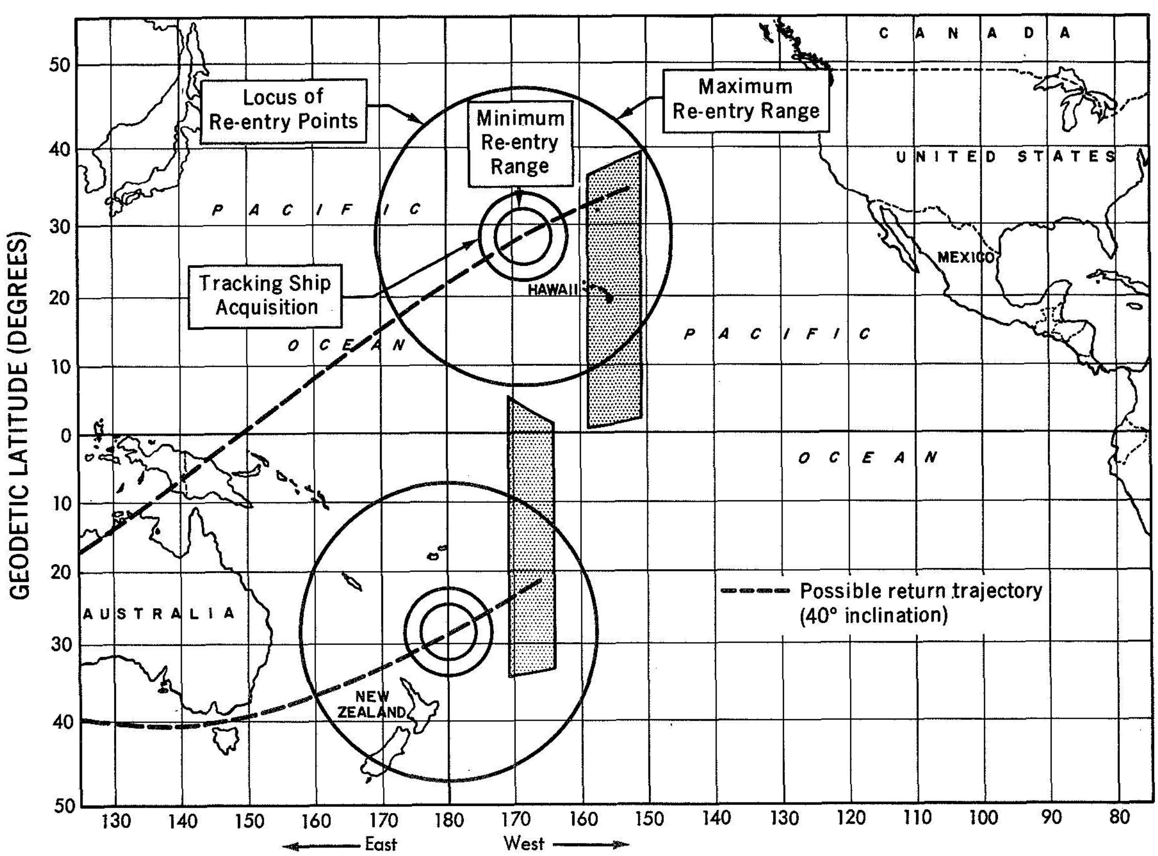

Figure 3-1—Water recovery areas

3.3 Reference Trajectories

Specific examples of complete missions are often desired for error analysis studies. For a lunar landing mission, two reference trajectories produced by the Mission Planning and Analysis Division are presented:

The May 6, 1968 mission

The September 17, 1969 mission

They are presented in patched conic and integrated trajectories form. The initial conditions for the various phases are given in tables found on the following 3 pages.

Table 3-1
INTEGRATED TRAJECTORY STATE VECTOR

May 6, 1968 Mission

Event	Hours from midnight of day of launch	x nm	y nm	z nm	$\dot{x}$ ft/sec	$\dot{y}$ ft/sec	$\dot{z}$ ft/sec
Earth Parking Orbit Insertion	18.725336	-125.24236E	+2985.4635E	+1899.2180E	-25526.544E	-1445.0171E	+588.20698E
Translunar Injection	21.394764	+3091.8028E	+1633.2175E	+883.53466E	-14646.307E	+27051.882E	+17921.139E
Lunar Orbit Insertion	87.915725	-1012.4370M	-51.263872M	-20.120039M	-287.89606M	+4914.2675M	+1967.3377M
LEM Hohmann Descent	91.830504	-896.94504M	-438.88333M	-175.67225M	-2430.6576M	+4280.7883M	+1714.2448M
LEM Hohmann Ascent	117.263463	+895.08756M	+290.59464M	+115.06502M	+1839.0358M	-4875.5506M	-1991.0614M
Transearth Injection	119.982536	-408.06592M	-865.54267M	-338.04017M	-7395.7897M	+2354.8007M	+1066.5033M

Earth centered coordinate system
x-y plane is earth equatorial plane
x points toward mean equinox of date of launch
z points north

Moon centered coordinate system
parallel to earth centered system but centered at moon

Table 3-2
INTEGRATED TRAJECTORY STATE VECTOR

September 17, 1969

Event	Hours from midnight of day of launch	x nm	y nm	z nm	$\dot{x}$ ft/sec	$\dot{y}$ ft/sec	$\dot{z}$ ft/sec
Earth Parking Orbit Insertion	13.019019	-2234.9353E	+1982.0656E	+1906.8056E	-16705.129E	-19364.552E	+548.50787E
Translunar Injection	15.920681	-1979.0874E	+2328.1528E	+1924.0770E	-27937.154E	-21544.653E	+4916.3660E
Lunar Orbit Insertion	77.170049	165.63936M	-919.37695M	-415.86151M	-5208.3665M	-817.04316M	-268.21399M
LEM Hohmann Descent	81.2345861	291.06763M	-892.26715M	-405.90489M	-4963.9006M	-1378.4721M	-526.42523M
LEM Hohmann Ascent	117.223599	-207.17281M	+842.58758M	+381.20313M	+5430.6187M	+1213.0308M	+459.61653M
Transearth Injection	121.986737	+952.74342M	-332.99696M	-166.43166M	-2725.8340M	-6776.7037M	-3294.6829M

Earth centered coordinate system
x-y plane is earth equatorial plane
x points toward mean equinox of date of launch
z points north

Moon centered coordinate system
parallel to earth centered system but centered at moon

Table 3-3
PATCHED CONIC, REFERENCE TRAJECTORY MAY 6, 1968 LUNAR LANDING MISSION

Coordinate Systems: Earth reference – geocentric; Moon reference – selenocentric
Units: Earth radii, earth radii per hour, hours, hours from base time, radians

Title t	x $\dot{x}$	y $\dot{y}$	z $\dot{z}$	r v	r^2 v^2	r^3 v^3
LAUNCH	4.39268231E-01	7.62512392E-01	4.74989402E-01	9.99998331E-01	9.99996662E-01	9.99994993E-01
-1.02311113E-01	-3.94712749E 00	1.52876310E 00	1.19612542E 00	4.39859617E 00	1.93476480E 01	8.51024890E 01
START EARTH PARK 0	3.87016323E-02	8.62274849E-01	5.60266262E-01	1.02903548E 00	1.05891401E 00	1.08966008E 00
9.56705511E-02	-4.39155066E 00	3.72288346E-02	2.46059209E-01	4.39859617E 00	1.93476480E 01	8.51024890E 01
END EARTH PARK OR.	1.01910535E-01	-9.03251374E-02	-1.10361424E-01	1.02903548E 00	1.05891401E 00	1.08966008E 00
2.54557133E 00	5.80552197E-01	3.66569167E 00	2.36078632E 00	4.39859617E 00	1.93476480E 01	8.51024890E 01
END TL INJECTION	1.00184192E 00	2.81375277E-01	1.32866684E-01	1.04905330E 00	1.10051282E 00	1.15449661E 00
2.63276577E 00	-1.79654628E 00	4.85828269E 00	3.25842994E 00	6.11946523E 00	3.74478543E 01	2.29160842E 02
PERIGEE TL CONIC	1.00184192E 00	2.81375277E-01	1.32866684E-01	1.04905331E 00	1.10051285E 00	1.15449664E 00
2.63276577E 00	-1.79654628E 00	4.85828269E 00	3.25842994E 00	6.11946523E 00	3.74478546E 01	2.29160848E 02
START LUNAR BRAKE	-2.92470470E-01	-4.06166226E-02	-1.61864783E-02	2.95720622E-01	8.74506867E-02	2.58609718E-02
6.90177828E 01	-2.09560582E-01	1.24362735E 00	6.65885401E-01	1.42615847E 00	2.03392798E 00	2.90070367E 00
FREE RETURN PERIGEE	1.00526671E 01	-1.10638066E-02	-3.69917974E-02	1.00600794E 00	1.01205198E 00	1.01813234E 00
1.91291772E 02	1.68171902E-01	5.53163821E 00	2.91605085E 00	6.25545007E 00	3.91306555E 01	2.44779861E 02
START LUNAR PARK 0	-2.92470387E-01	-4.06166112E-02	-1.61864738E-02	2.95720625E-01	8.74506867E-02	2.58609718E-02
6.90177828E 01	-1.34545290E-01	8.35652864E-01	3.34175667E-01	9.09995353E-01	8.28091538E-01	7.53559446E-01
END LUNAR PARK OR.	-9.32257295E-02	-2.60587010E-01	-1.04182822E-01	2.95720625E-01	8.74506867E-02	2.58609718E-02
1.01329254E 02	-8.63593614E-01	2.66313663E-01	1.06651974E-01	9.09995353E-01	8.28091538E-01	7.53559446E-01
END EARTH INJEC.	-9.32257295E-02	-2.60587010E-01	-1.04182822E-01	2.95720622E-01	8.74506867E-02	2.58609715E-02
1.01329254E 02	-1.28074151E 00	3.27462155E-01	3.26980367E-01	1.36178061E 00	1.85444644E 00	2.52534920E 00
RETURN PERIGEE	9.14802229E-01	3.92476043E-01	1.73541676E-01	1.01045400E 00	1.02101728E 00	1.03169100E 00
1.94656856E 02	-2.55255467E 00	4.32308584E 00	3.67815006E 00	6.22361588E 00	3.87333944E 01	2.41061771E 02
START REENTRY	9.84092081E-01	2.57890725E-01	6.04577744E-02	1.01911727E 00	1.03860000E 00	1.05845520E 00
1.94626440E 02	-1.99927737E 00	4.51277888E 00	3.74648130E 00	6.19664472E 00	3.83984053E 01	2.37941280E 02
DURING REENTRY	9.84092081E-01	2.57890725E-01	6.04577744E-02	1.01911727E 00	1.03860000E 00	1.05845518E 00
1.94757727E 02	-1.04032524E 00	3.33505982E 00	2.70758969E 00	4.41994828E 00	1.95359427E 01	8.63478553E 01

Title E P_x/p	a M P_y/p	e n P_z/p	i T $r_{Periapsis}$	Ω H_x/h $v_{Periapsis}$	ω H_y/h $\bar{R}_o \cdot \dot{\bar{R}}_o = d_o$	t_p H_z/h $1/a$
TE CONIC EARTH REF	3.91097167E 01	9.73176646E-01	5.79163361E-01	7.73013914E-02	1.80427995E 00	3.47044227E 02
3.02156869E-06	2.59154615E-05	1.82432520E-02	3.44411469E 02	4.22667783E-02	-5.45689481E-01	8.36920846E-01
4.09255892E-01	-7.54712254E-01	-5.12756324E-01	1.04905359E 00	6.11946428E 00	8.20532441E-05	2.55690929E-02
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1.72633491E-04	4.00583965E-04	9.48064423E-01	6.62738216E 00	-1.63987185E-02	4.69819701E-01	-8.82610059E-01
-9.89009368E-01	-1.37347953E-01	-5.47357094E-02	2.95720613E-01	1.42615849E 00	0.	-1.54254758E 00
FR CONIC EARTH REF	4.42039412E 01	9.77241647E-01	4.86584496E-01	5.86068481E-02	2.84586555E 00	1.31291772E 02
-1.74707893E 00	-2.96234751E 00	1.51823069E-02	4.13849175E 02	2.73894370E-02	-4.66806728E-01	8.83935118E-01
9.99548912E-01	2.36881304E-02	-1.84621055E-02	1.00600876E 00	6.25544566E 00	-2.85416135E 01	2.26224166E-02
TE CONIC MOON REF	-1.23514409E 00	1.23942195E 00	2.70114407E 00	-2.83944327E 00	5.98884910E-01	1.01329394E 02
-2.11431983E-04	-6.46631187E-04	3.60498571E-01	1.74291542E 01	-1.26868799E-01	4.07031429E-01	-9.04560506E-01
-3.15249586E-01	-8.81193161E-01	-3.52301449E-01	2.95720601E-01	1.36178064E 00	-5.96046448E-08	-8.09622121E-01
TE CONIC EARTH REF	2.95619673E 01	9.65819108E-01	6.62956524E-01	1.80115651E-01	2.90386266E 00	1.94656856E 02
-2.65440521E 00	-3.07607299E 00	2.77606264E-02	2.26334417E 02	1.10253739E-01	-6.05493671E-01	7.88176072E-01
7.39568490E-01	5.79760504E-01	3.41930106E-01	1.01045427E 00	6.22361487E 00	-1.09690680E 01	3.38277473E-02

where $\bar{H} = (H_x, H_y, H_z) = \bar{R} \times \dot{\bar{R}}$

$\bar{P} = (P_x, P_y, P_z)$ = vector in direction of periapsis

Table 3-4
PATCHED CONIC, REFERENCE TRAJECTORY SEPTEMBER 17, 1969 LUNAR LANDING MISSION

Coordinate Systems: Earth reference – geocentric; Moon reference – selenocentric
Units: Earth radii, earth radii per hour, hours, hours from base time, radians

Title	x	y	z	r	r^2	r^3
t	$\dot{x}$	$\dot{y}$	$\dot{z}$	v	v^2	v^3
LAUNCH	-2.61907521E-01	8.38973331E-01	4.76986212E-01	9.99993825E-01	9.99987662E-01	9.99981487E-01
1.27145 4E-02	-3.80045062E 00	-1.86518437E 00	1.19389898E 00	4.39860576E 00	1.93477324E 01	8.51030469E 01
START EARTH PARK 0	-6.06858200E-01	6.12262309E-01	5.61927658E-01	1.02903098E 00	1.05890475E 00	1.08964579E 00
2.10492373E-01	-3.00693446E 00	-3.20127439E 00	2.40666047E-01	4.39860576E 00	1.93477324E 01	8.51030469E 01
END EARTH PARK OR.	-1.99110475E-01	8.94833708E-01	4.67474461E-01	1.02903098E 00	1.05890475E 00	1.08964579E 00
2.03425193E 00	-3.87894809E 00	-1.57067959E 00	1.35442238E 00	4.39860576E 00	1.93477324E 01	8.51080469E 01
END TL INJECTION	-5.73888361E-01	6.75075561E-01	5.61630416E-01	1.04904890E 00	1.10050359E 00	1.15448208E 00
3.02145195E 00	-4.81025964E 00	-3.69884905E 00	8.46055424E-01	6.12665421E 00	3.75358918E 01	2.29969433E 02
PERIGEE TL CONIC	-3.77181831E-01	8.11086261E-01	5.20740676E-01	1.03503522E 00	1.07129793E 00	1.10883111E 00
2.08200706E 00	-5.14285642E 00	-3.17808247E 00	1.22500001E 00	6.16845244E 00	3.80498055E 01	2.34708416E 02
START LUNAR BRAKE	9.14372098E-02	-2.55935153E-01	-1.16563806E-01	2.95720625E-01	8.74506867E-02	2.58609718E-02
6.46105701E 01	-1.35900301E 00	-2.95845199E-01	-4.16477954E-01	1.451849 E 00	2.10786745E 00	3.06030664E 00
FREE RETURN PERIGEE	-1.87690558E-01	9.22975099E-01	3.53934580E-01	1.00617120E 00	1.01238048E 00	1.01862808E 00
1.32938558E 02	-3.51597363E-01	-2.29643866E 00	5.80210078E 00	6.24992996E 00	3.90616247E 01	2.44132420E 02
START LUNAR PARK 0	9.14372289E-02	-2.55935207E-01	-1.16563828E-01	2.95720625E-01	8.74506867E-02	2.58609718E-02
6.46105701E 01	-8.65258348E-01	-2.62905964E-01	-1.01488878E-01	9.09995353E-01	8.28091538E-01	7.53559446E-01
END LUNAR PARK OR.	2.82698664E-01	-7.70484251E-02	-3.99461126E-02	2.95720625E-01	8.74506867E-02	2.58609718E-02
1.11261301E 02	-2.66597763E-01	-7.95718211E-01	-3.51922849E-01	9.09995353E-01	8.28091538E-01	7.53559446E-01
END TEARTH INJEC.	2.82698664E-01	-7.70484251E-02	-3.99461126E-02	2.95720625E-01	8.74506867E-02	2.58609718E-02
1.11261301E 02	-4.00613683E-01	-1.18597727E 00	-5.47617799E-01	1.36635232E 00	1.86691865E 00	2.55086863E 00
RETURN PERIGEE	-5.84317166E-01	7.16488701E-01	3.95784074E-01	1.00569759E 00	1.01142766E 00	1.01719037E 00
1.99359500E 02	-4.92805803E 00	-2.34832311E 00	-3.02438161E 00	6.24077410E 00	3.89472610E 01	2.43061060E 02
START REENTRY	-3.92325053E-01	7.94397622E-01	5.03314632E-01	1.01897603E 00	1.03831215E 00	1.05801520E 00
1.99321988E 02	-5.28316671E 00	-1.79985528E 00	-2.69814485E 00	6.19929957E 00	3.84313151E 01	2.38247237E 02
DURING REENTRY	-3.92325053E-01	7.94397622E-01	5.03314632E-01	1.01897603E 00	1.03831215E 00	1.05801518E 00
1.99478935E 02	-3.98530576E 00	-8.99016738E-01	-1.68752992E 00	4.42025459E 00	1.95386504E 01	8.63658082E 01

Title	a	e	i	Ω	ω	t_p
E	M	n	T	H_x/h	H_y/h	H_z/h
P_x/p	P_y/p	P_z/p	$r_{Periapsis}$	$v_{Periapsis}$	$\bar{R}_o \cdot \dot{\bar{R}}_o = d_o$	$1/a$
TL CONIC EARTH REF	4.72697008E 01	9.78103614E-01	5.71495003E-01	8.74071002E-01	2.76564267E 00	2.98200703E 00
9.68387449E-09	9.20423007E-08	1.37295079E-02	4.57640964E 02	4.14834416E-01	-3.47094378E-01	8.41093278E-01
-3.64414412E-01	7.83631605E-01	5.03113943E-01	1.03503557E 00	6.16845137E 00	2.90572643E-07	2.11552000E-02
TL CONIC MOON REF	-5.42156887E-01	1.54545210E 00	2.63236672E 00	2.78988144E 00	-6.28742081E-01	6.46106458E 01
-1.72633491E-04	-3.72931626E-04	1.23963125E 00	5.06859219E 00	1.67946595E-01	4.57658923E-01	-8.73122132E-01
3.09201324E-01	-8.65462637E-01	-3.94168675E-01	2.95720613E-01	1.45184967E 00	-4.65661287E-10	-1.84448452E-01
ER CONIC EARTH REF	3.88108078E 01	9.74074960E-01	1.45035979E 00	1.72592118E 00	1.93299429E 00	1.32938558E 02
-2.20131457E-09	-1.92089786E-08	1.84544128E-02	3.40470609E 02	9.80835509E-01	1.53384295E-01	1.20145589E-01
-1.86539386E-01	9.17314172E-01	3.51763794E-01	1.00617146E 00	6.24992901E 00	-5.96046448E-08	2.57660180E-02
TE CONIC MOON REF	-1.16204329E 00	1.25448330E 00	2.70490575E 00	-3.11126426E 00	-1.24520184E 00	1.11261413E 02
-1.72633491E-04	-5.13829362E-04	3.95044962E-01	1.59049876E 01	-1.28251018E-02	4.22745109E-01	-9.06157863E-01
9.55965233E-01	-2.60544917E-01	-1.35080704E-01	2.95720607E-01	1.36635233E 00	-7.12461770E-08	-8.60553133E-01
TE CONIC EARTH REF	3.08202034E 01	9.67368877E-01	6.74212325E-01	-3.21336824E-01	-2.25285298E 00	1.99359500E 02
-5.59658986E-09	-4.34560633E-08	2.60781038E-02	2.40937197E 02	-1.97170329E-01	-5.92327696E-01	7.81198919E-01
-5.81006861E-01	7.12429553E-01	3.93541813E-01	1.00569780E 00	6.24077338E 00	-1.34110451E-07	3.24462482E-02

where $\bar{H} = (H_x, H_y, H_z) = \bar{R} \times \dot{\bar{R}}$

$\bar{P} = (P_x, P_y, P_z)$ = vector in direction of periapsis

4.0 TRAJECTORY PREDICTION CAPABILITY

4.1 The trajectory prediction model is of interest since it influences the accuracy with which the orbit may be computed at some initial time and future times. The error in the computed orbit due to the predictor may be determined by investigating the influence of the error in gravitational constants, atmospheric drag, S-IV-B venting, etc.

The following presents the values and associated uncertainties of the constants to be used in trajectory prediction model. Also presented is a drag model, a venting model, the earth and moon potential model, and a designation of the celestial body ephemeris system to be used. A list of conversion factors and a description of potential equations are also included.

Uncertainties are presented, if available. For some quantities, more decimal digits are given than are justified by the uncertainties, for purposes of consistent conversion between units. The values presented are in agreement with those adopted by NASA Headquarters and with those proposed by W. Kaula and V. Clarke. These documents are referenced at the end of the chapter.

4.1.1 Earth Constants for Apollo

Equatorial earth radius (gravitational)

One earth radius = 6.378 165 (±0.000 025) × 10^6 m (reference b)
One earth radius = 2.092 573 819 (±0.000 008 2) × 10^7 international feet
One earth radius = 3.443 933 585 (±0.000 013 5) × 10^3 nautical miles
One earth radius = 3.963 207 990 530 (±0.000 015 5) × 10^3 international statute miles
One earth radius = 3.963 200 063 920 (±0.000 015 5) × 10^3 U. S. statute miles

Flattening

f = 1/298.30 (accepted number based on 1960 Fischer Ellipsoid)
f = 3.352 329 869 259 14 × 10^{-3}

Gravitational parameter ($G M_e = \mu_e = \mu$ earth)

μ earth = 3.986 032 ($\pm$0.000 030) $\times 10^{14}$ m^3/sec^2 (reference b)
μ earth = 1.407 653 92 ($\pm$0.000 010 6) $\times 10^{16}$ (international feet)^3/sec^2
μ earth = 1.990 941 65 ($\pm$0.000 015) $\times$ 10 (earth radii)^3/hour^2

Mass of the earth

$M_e = 5.975 \times 10^{24}$ kg (reference b)

Angular velocity of the earth rotation

The equation which was taken from reference b follows:

$$\omega = \frac{2\pi}{86\ 164.098904 + 0.001\ 64T} \text{ radians/sec.}$$

Where T is the number of Julian centuries of 36525 days from 1900 January 0.5 U.T. (Julian date = 241 502 0.0).

$\omega = 7.292\ 156\ 06 \times 10^{-5}$ radians/second (1966-1972)

$\omega = 2.625\ 161\ 42$ radians/hour

$\omega = 4.178\ 074\ 16 \times 10^{-3}$ degrees/second

Gravitational potential function

$$U = \frac{\mu_e}{r}\left[1 - \frac{J_2 R_e^2}{2 r^2}(3 \sin^2 \phi - 1) - \frac{J_3 R_e^3}{2 r^3}(5 \sin^3 \phi - 3 \sin \phi)\right.$$

$$- \frac{J_4 R_e^4}{8 r^4}(35 \sin^4 \phi - 30 \sin^2 \phi + 3)$$

$$\left. + 3 J_{22} \frac{R_e^2}{r^2} \cos^2 \phi \cos^2 (\lambda - \lambda_{22})\right]$$

$J_2 = 1082.3\ (\pm 0.2) \times 10^{-6}$ (reference b)

$J_3 = -2.3\ (\pm 0.1) \times 10^{-6}$ (reference b)

$J_4 = -1.8\ (\pm 0.2) \times 10^{-6}$ (reference b)

$\lambda_{22} = -21.°0\ (\pm 3.°0)$

$J_{22} = 1.9\ (\pm .2) \times 10^{-6}$

r = distance from the center of the earth to the vehicle

ϕ = declination of the vehicle

$R_e = 6.378\ 165 \times 10^6$ m

λ = longitude of vehicle

4.1.2 Lunar Constants

Earth-moon mass ratio (reference b)

$M_e/M_m = 81.3015\ (\pm 0.0033)$

Mean lunar radius (reference b)

$R_m = 1.738\ 09 \times 10^6$ m

$R_m = 5.702\ 395 \times 10^6$ international feet

$R_m = 2.725\ 06 \times 10^{-1}$ earth radii

Moments of inertia about principal rotational axes

$A = 0.887\ 817\ 983\ 4 \times 10^{35}$ kg meters^2

$B = 0.888\ 001\ 954\ 2 \times 10^{35}$ kg meters^2

$C = 0.888\ 369\ 781\ 7 \times 10^{35}$ kg meters^2

Principal axes (reference b)

$a = 1.738\ 57 \times 10^6$ m

$b = 1.738\ 21 \times 10^6$ m

$c = 1.737\ 49 \times 10^6$ m

where a is directed toward the center of the earth, c is coincident with the moon's rotational axis, and b is perpendicular to a and c

Gravitational potential functions

$$U = \frac{\mu_m}{r}\left[1 - \frac{J_2 R_m^2}{2 r^2} (3 \sin^2 \phi - 1) + \frac{3J_{22} R_m^2}{r^2} \cos^2 \phi \cos 2T\right]$$

$J_2 = 2.071\ 08\ 10^{-4}$

$J_{22} = 2.071\ 60\ 10^{-5}$

ϕ = selenographic declination of the vehicle

T = selenographic right ascension of the vehicle

r = distance from the center of the moon to the vehicle

$R_m = 1.73809 \times 10^6$ m

Rotational rate of the moon (from reference e)

$$\omega_m = \frac{2\pi}{2360\ 591.545 + 0.014\ T} \text{ radians/second}$$

Where T is the number of Julian centuries of 36525 days from 1900 January 0.5 U.T.

$\omega_m = 2.661\ 699\ 477 \times 10^{-6}$ radians/second (1968-1970)

$\omega_m = 9.582\ 118\ 118 \times 10^{-3}$ radians/hour

$\omega_m = 1.525\ 041\ 464 \times 10^{-4}$ degrees/second

Earth Radii/Kilometers Conversion Factor for the Lunar Ephemeris

$b = 86.315\ 745\ (GM_e + GM_m)^{\frac{1}{3}}$

$b = 6.378\ 325\ 5 \times 10^6$ m (reference b)

Gravitational parameters for the moon ($GM_m = \mu_m = \mu$ moon)

μ moon = $4.902\ 778 \times 10^{12}$ m^3/sec^2 (reference b)

μ moon = $1.731\ 399\ 71 \times 10^{14}$ (international feet)^3/sec^2

μ moon = $2.448\ 837\ 57 \times 10^{-1}$ (earth radii)^3/hours^2

Mean distance of moon with respect to earth

$3.844\ 020 \times 10^{8}$ m

$2.388\ 56 \times 10^{5}$ international statute miles

$6.026\ 84 \times 10^{1}$ earth radii

$2.075\ 60 \times 10^{5}$ nautical miles

4.1.3 General Constants

Astronomical unit (reference b)

$AU = 1.495\ 990\ 00 \times 10^{11}$ m

Velocity of light in a vacuum

$c = 2.997\ 925\ (\pm 0.000\ 001) \times 10^{8}$ m/sec (reference b)

$c = 9.835\ 711\ (\pm 0.000\ 0033) \times 10^{8}$ international feet/sec

$c = 4.700\ 294\ (\pm 0.000\ 0016) \times 10$ earth radii/sec

$c = 1.862\ 824\ (0.000\ 000\ 62) \times 10^{5}$ international miles/sec

Gravitational parameters for the sun

μ sun $= 1.327\ 154\ 45 \times 10^{20}$ m^3/sec^2 (reference b)

μ sun $= 4.686\ 801\ 71 \times 10^{21}$ (international feet)^3/sec^2

μ sun $= 6.628\ 865\ 68 \times 10^{6}$ (earth radii)^3/hour^2

4.1.4 Ephemeris Tape System

The ephemeris tape system to be used for Apollo missions is provided by JPL and is called the JPL Ephemeris Tapes. For additional information, see reference g.

A one sigma for the radius vector of the moon with respect to the earth is one kilometer.

The value of the earth radius = 6378.3255 km is used for converting the lunar ephemeris from the earth radii of the ephemeris tape to the units used in the users' program.

4.1.5 Drag Model

Atmospheric Model for Apollo

The 1963 Patrick Reference Atmosphere is to be used for the lower altitudes during the launch phase. The 1962 U. S. Standard Atmosphere is to be used for the higher altitudes and reentry. This atmosphere is described in MSFC memorandum R-AERO Y-12-63.

Drag equations for the parking orbit phase

$$\ddot{R} = -\frac{1}{2} C_D \, \rho \, (A/M) \, (\dot{R} - \Omega \times R) \, |\dot{R} - \Omega \times R|$$

or in component form:

$$\ddot{x} = -\frac{1}{2} C_D \, \rho \, (A/M)(\dot{x} + \Omega y) \; v$$

$$\ddot{y} = -\frac{1}{2} C_D \, \rho \, (A/M) \, (\dot{y} - \Omega x) \, v$$

$$\ddot{z} = -\frac{1}{2} C_D \, \rho \, (A/M) \, (\dot{z}) \, v$$

$$v = [(\dot{x} + \Omega y)^2 + (\dot{y} - \Omega x)^2 + \dot{z}^2]^{\frac{1}{2}}$$

where

$\ddot{x}_D$, $\ddot{y}_D$, $\ddot{z}_D$ = components of acceleration due to drag

x, y, z = inertial position components of vehicle

C_D = drag coefficient

A = frontal surface area of vehicle

M = mass of vehicle

ρ = density of atmosphere at given altitude (z_1)

$\dot{R}$ = inertial velocity vector of vehicle

R = position vector to vehicle from the center of earth

$\ddot{R}$ = acceleration of vehicle

Ω = earth's rotational vector

v = speed of vehicle with respect to rotating earth

The following values have been adopted for the drag model.

Drag coefficient

$C_D = 2.0 \pm 0.2$

Area

$A = 129.9 \text{ feet}^2 \; (12.1 \text{ m}^2)$

Mass

M = 280,000 lb (127 000 kg) (parking orbit)

M = 130,000 lb (59 000 kg) (after translunar thrust)

M = 9,500 lb (4300 kg) (reentry)

4.1.6 Fischer Earth Model

The following constants describe the Fischer earth model (1960) which is used for the location of radar stations for Apollo and Gemini (reference d and f).

Equatorial earth radius

$a = 6.378\ 166\ 000 \times 10^6$ meters (exact)

$a = 2.092\ 574\ 147 \times 10^7$ international feet

Flattening

f = flattening $= 1 - b/a$

$f = 1/298.30$

Polar earth radius

$b = 6.356\ 784\ 284 \times 10^{6}$ meters

$b = 2.085\ 559\ 148 \times 10^{7}$ international feet

Eccentricity of ellipsoid

$$e = \sqrt{\frac{a^2 - b^2}{a^2}}$$

$e = 8.181\ 333\ 402 \times 10^{-2}$

$e^2 = 2f - f^2$

$e^2 = 6.693\ 421\ 623 \times 10^{-3}$

4.1.7 Basic Equivalents and Conversion Factors

1 international foot = 3.048×10^{-1} (exact) meters (reference b)

1 nautical mile = 1.852 (exact) kilometers (reference b)

1 meter = 3.280 839 895 013 12 international feet

1 meter^2 = $1.076\ 391\ 041\ 670\ 97 \times 10$ (international feet)^2

1 nautical mile = $6.076\ 115\ 485\ 564\ 30 \times 10^{3}$ international feet

1 U. S. foot = $3.048\ 006\ 096\ 012\ 192\ 0 \times 10^{-1}$ meters

1 international statute mile = 5.280×10^{3} (exact) international feet

1 U. S. statute mile = 5.280×10^{3} (exact) U. S. feet

π = 3.141 592 653 589 793 238 47 (reference c)

1 degree = $1.745\ 329\ 251\ 994\ 329\ 577 \times 10^{-2}$ radians

1 radian = $5.725\ 577\ 951\ 308\ 232\ 1 \times 10$ degrees

1 lbm = $4.535\ 923\ 7 \times 10^{-1}$ kg (exact)

1 kg = 2.204 622 621 848 78 lbm

1 lbf = 1 lbm .32.174 048 556 feet/sec^2

1 kilometer = $1.567\ 849\ 060\ 035\ 292 \times 10^{-4}$ earth radii

1 earth radius = $6.378\ 165\ 0 \times 10^{3}$ kilometers (exact for scaling)

1 kilometer/sec = 5.644 256 616 127 052 $\times 10^{-1}$ earth radii/hour

1 earth radius/hour = 1.771 712 5 (exact) kilometers/sec

1 earth radius/hour = 5.812 705 052 493 438 $\times 10^{-3}$ international feet/sec

1 international foot/sec = 1.720 369 416 595 526 $\times 10^{-4}$ earth radii/hour

1 kilometer = 5.399 568 034 557 235 42 $\times 10^{-1}$ nautical miles

4.1.8 Clarification of the Potential Equations

The following is a clarification of section 4.1.1 which gave the gravitation potential function for the earth. The four forms in use follow the associated constants.

$$U = \frac{\mu_e}{r}\left[1 - \frac{J_2}{2}\left(\frac{R_e}{r}\right)^2 (3 \sin^2 \phi - 1) - \frac{J_3}{2}\left(\frac{R_e}{r}\right)^3 (5 \sin^3 \phi - 3 \sin \phi) - \frac{J_4}{8}\left(\frac{R_e}{r}\right)^4 (35 \sin^4 \phi - 30 \sin^2 \phi + 3) + 3 J_{22}\left(\frac{R_f}{r}\right)^2 \cos^2 \phi \cos 2 (\lambda - \lambda_{22})\right]$$

$$U = \frac{\mu_e}{r}\left[1 + \frac{C_{2,0}}{2}\left(\frac{R_e}{r}\right)^2 (3 \sin^2 \phi - 1) + \frac{C_{3,0}}{2}\left(\frac{R_e}{r}\right)^3 (5 \sin^3 \phi - 3 \sin \phi) + \frac{C_{4,0}}{8}\left(\frac{R_e}{r}\right)^4 (35 \sin^4 \phi - 30 \sin^2 \phi + 3) + 3 J_{22}\left(\frac{R_f}{r}\right)^2 \cos^2 \phi \cos 2 (\lambda - \lambda_{22})\right]$$

$$U = \frac{\mu_e}{r}\left[1 + \frac{J}{3}\left(\frac{R_e}{r}\right)^2 (1 - 3\sin^2\phi)\right.$$

$$+ \frac{H}{5}\left(\frac{R_e}{r}\right)^3 (3\sin\phi - 5\sin^3\phi)$$

$$+ \frac{D}{35}\left(\frac{R_e}{r}\right)^4 (3 - 30\sin^2\phi + 35\sin^4\phi)$$

$$\left. + 3J_{22}\left(\frac{R_f}{r}\right)^3 \cos^2\phi \cos 2(\lambda - \lambda_{22})\right]$$

The fourth order term of the previous equation is sometimes written as follows:

$$+ \frac{K}{30}\left(\frac{R_e}{r}\right)^4 (3 - 30\sin^2\phi + 35\sin^4\phi)$$

The associated definitions and constants are:

μ_e = gravitational parameter for the earth

R_e = equatorial earth radius (gravitational)

$J = 1.62345\ (\pm 0.00030) \times 10^{-3}$

$H = -0.575\ (\pm 0.025) \times 10^{-5}$

$D = 0.7875\ (\pm 0.0876) \times 10^{-5}$

$K = 6.750 \times 10^{-6}$

$C_{2,0} = -1\ 082.30 \times 10^{-6}$

$C_{3,0} = 2.3 \times 10^{-6}$

$C_{4,0} = 1.8 \times 10^{-6}$

Relations between the above are as follows:

$$J_2 = \frac{2}{3}J,\quad J_3 = \frac{2}{5}H,\quad J_4 = \frac{-4}{15}K = -\frac{8}{35}D$$

$$J_2 = -C_{2,0},\quad J_3 = -C_{3,0},\quad J_4 = -C_{4,0}$$

4.1.8 This is a clarification of section 4.1.2, Gravitational Potential Functions for the moon. The three forms of the function follow:

$$U = \frac{\mu_m}{r}\left[1 - \frac{J_2 R_m^2}{2r^2}(3 \sin^2 \phi - 1) + \frac{3 J_{22} R_m^2}{r^2} \cos^2 \phi \cos 2T\right]$$

Other forms are:

$$U = \frac{\mu_m}{r}\left[1 + \frac{J}{3}\left(\frac{R_m}{r}\right)^2 (1 - 3 \sin^2 \phi) + L\left(\frac{R_m}{r}\right)^2 \cos^2 \phi \cos 2T\right]$$

$J = 3106.6 \times 10^{-7}$

$L = 621.48 \times 10^{-7}$

$L = 3 J_{22}$ and $J = \frac{3}{2} J_2$

$$U = \frac{\mu_m}{r}\left[1 + \frac{C_{2,0}}{2}\left(\frac{R_m}{r}\right)^2 (3 \sin^2 \phi - 1)\right.$$

$$\left. + 3 C_{2,2}\left(\frac{R_m}{r}\right)^2 \cos^2 \phi \cos 2T\right]$$

$C_{2,0} = - J_2$ and $C_{2,2} = J_{22}$

4.2 Venting Model

4.2.1 Earth Orbit

Low level continuous vents of approximately 6.7 lbf (3.0 kgf) ±10 percent directed parallel to the longitudinal axis of the vehicle are planned. The direction of the vent is such to increase the speed. These vents could increase to 11.6 lbf (5.3 kgf) ±10 percent or decrease to 0.0 lbf (0 kgf) if unanticipated heating or cooling conditions are encountered.

4.2.2 Translunar Trajectory

Immediately after translunar injection, a nonpropulsive venting blowdown will be initiated, which should last 100 seconds. The nonpropulsive vent is to be achieved by the use of opposing vent ports to null each other's effect. The one sigma uncertainty in the velocity associated with the nonpropulsive vent process is 3 feet/second (1 m/sec). There will be no further venting for a one hour period succeeding the blowdown.

4.2.3 The above information represents the current thinking of Marshall Space Flight Center and was presented at the Tenth Flight Mechanics Panel meeting held during 22-23 October 1964 at the Manned Spacecraft Center.

REFERENCES

a. Technical Note D-1847, "A review of Geodetic Parameters," William M. Kaula, Goddard Space Flight Center, Greenbelt, Maryland, May, 1963.

b. Technical Report No. 32-604, "Constants and Related Data for Use in Trajectory Calculations," Victor C. Clarke, Jr., California Institute of Technology, Pasadena, California, March 6, 1964.

c. "Standard Mathematical Tables," Twelfth Edition, Chemical Rubber Publishing Company, Cleveland, Ohio, 1959.

d. NASA M-DE-8020-008, "Natural Environment and Physical Standards for Project Apollo," April 1, 1963.

e. "Standard Astrodynamic Constants and Conversion Factors for Project Apollo," J. O. Cappellari, Jr., Bellcomm, Inc., Washington, D. C., January 8, 1964.

f. "Geophysical Constants and Geodetic Coordinates for Use in Gemini Planning Studies," Paul G. Brumberg, Manned Spacecraft Center, Houston, Texas, January 18, 1963.

g. Technical Report 32-580, "Users' Description of Jet Propulsion Laboratory Ephemeris Tapes," P. Peabody, J. F. Scott, and E. G. Orozco, Jet Propulsion Laboratory, Pasadena, California, March 2, 1964.

5.0 MEASUREMENT AND COMMUNICATION SYSTEMS

Several measuring systems exist to determine the vehicle position and velocity. These are classified as being either onboard or ground based. Certain characteristics of these systems such as measurement acquisition limitations, accuracy, systematic errors, and frequency of measurement are key factors in error analysis studies involving orbit determination.

5.1 Ground Based Measuring Devices

5.1.1 The characteristics of the ground based measuring systems, the C-band radars and unified S-band system, are given in Tables 5-1 through 5-3.* In Table 5-1, the specific types of tracking equipment are listed, together with their characteristic signal bands and antenna mounts. The maximum tracking rate and angular rotation, the acquisition times, and the range limits are all limitations upon the radar system in acquiring a measurement. The figures in the "operational accuracy" columns of Table 5-1 represent current best estimates (cf. references 5-a through 5-e) of the standard deviations of the noise and bias errors to be expected in data reported from stations equipped with the listed tracking devices. Noise, as used in this report, will be understood as the zero-mean, Gaussian distributed errors of short correlation time (less than one second). Bias, as used in this report, means the zero-mean, Gaussian distributed errors of longer correlation times (approximately one hour). The noise and bias errors, combined with the station location error, astronomical constant errors, refractivity correction errors and antenna mount correction errors should yield the expected RMS deviation of the observed measurements. It has been agreed to adopt a figure which is twice the corresponding noise figure to represent unknown biases for purposes of error analysis, until better figures become available.

5.1.2 The information in Tables 5-2 and 5-3 present values for the standard deviations of the errors in the knowledge of the location and orientation of the physical antenna mount in a theoretical topocentric coordinate system, and the error in gauging the refractive index.

5.1.3 Table 5-2 presents the actual sites comprising the MSFN to the best of present knowledge. This data will be updated and revised as the planned network becomes implemented. Operational dates reflect the present scheduling for the new sites. Considered sites should be used only in an attempt to show their value.

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*Tables appear on pages 34 – 37

5.1.4 The location of the stations is given in the following two coordinate systems:

Geodetic Coordinates: The geodetic coordinates of the existing sites are referenced to a common datum, designated as the Astrogeodetic World Datum (Fischer ellipsoid, 1960) which was established as the NASA reference model for the Apollo missions. Height above the ellipsoid is the sum of the elevation above mean sea level (geoid) and the geoidal height. The standard deviations for the coordinates are based on uncertainties in the geodetic surveys.

Earth Fixed Geocentric Rectangular Coordinates: The components are expressed in meters of a rectangular coordinate system which rotates with the earth. The U axis is defined as being positive toward zero latitude and longitude, the V axis positive toward the equator at 90° east longitude with the W axis positive along the earth's polar axis. The standard deviations for the coordinates are given.

5.1.5 The locations and uncertainties in the geodetic coordinates used based upon surveying techniques are those quoted by Geonautics in Goddard Directory of Tracking Station Locations. They do not at this time reflect the location accuracies which might be achieved by satellite tracking techniques.

5.1.6 The geodetic data information for each station refers to the existing radar (or planned radar if none exists). If USBS or C-band radars are scheduled for implementation at a station where a radar now exists, the geodetic data for the existing site is implied.

5.1.7 In Table 5-3 the corrections relating the actual antenna mount and a true topocentric system are listed for the known systems, along with their uncertainties estimated for all systems. The uncertainty to be expected in a refractivity estimate is also listed, based upon the possible variation of the actual refractivity from the seasonal estimate. This is an average uncertainty since some areas experience more variation than others. The actual corrections are added to the angles and slant range in the following manner:

$$L_1 = L_1 + \Delta_{DV_1} + \Delta_{B_1} + \Delta_{R_1} \quad \text{where} \quad L_1 = \text{angle one}$$

$$L_2 = L_2 + \Delta_{DV_2} + \Delta_{B_2} + \Delta_{R_2} \qquad L_2 = \text{angle two}$$

$$r = r + \Delta r_R \qquad r = \text{slant range}$$

5.1.8 Δ_{DV_1} and Δ_{DV_2} are two components in the correction for the deflection in the meridian and prime vertical. These components are measured positive when the astronomical latitude and longitude are north and east of the geodetic position.

5.1.9 Δ_{B_1} and Δ_{B_2} are two components of the boresight correction. The first boresight angle correction is in the clockwise angle measured from true north to the boresight tower. The second angle correction is measured positive above the azimuth plane to the boresight.

5.1.10 Δ_{R_1}, and Δ_{R_2}, and Δ_{r_R} are the three components of the refractivity correction. It takes into account the bending of the signal while passing from the atmosphere to free space.

5.1.11 Two equipment independent error sources not specified in the tables are 1) the error, one part in 10^6, in the speed of light and 2) the difference in basic clock frequency between any two USBS stations. This difference is approximately constant with a value of one part in 10^{10} plus a random component addition of five parts in 10^{11}.

5.2 Onboard Based Measurements

5.2.1 The onboard based measurement system consists of a sextant and a telescope mounted upon an inertial platform. Observations are made by looking at a set of inertially located landmarks or stars.

5.2.2 The number of landmarks and their uncertainty is a large factor in the accuracy of the onboard navigation system.

5.2.3 Presented in the following table are ten lunar landmarks that have been recommended by the Lunar Surface Technology Branch (LSTB) of the Manned Spacecraft Center for optical sighting. They were taken from a list of 62 control points that were given to the LSTB by the Army Map Service. Thirty-eight of these control points are considered to be suitable for landmark identification points. Of these 38, the ten that have been chosen for navigation purposes lie within approximately ±5° of the lunar equator.

5.2.4 There is a great deal of controversy about the uncertainties associated with lunar landmarks. On the pessimistic side, uncertainties of lunar landmarks determined by studies conducted by the Air Force Aeronautical Chart and Information Center increase as the landmark's great circle distance increases from 0° latitude and 0° longitude. At the (0°, 0°) point, the probable errors are 2.74 nm (5070 meters) horizontal and .74 nm (1370 meters) vertical.

5.2.5 On the optimistic side, MIT and Army Map Service give lunar landmark uncertainties of .59 nm (1094 meters) horizontal and .46 nm (858) meters vertical, which are constant over a very large portion of the front side of the moon. In either case, it is hoped that successful Lunar Orbiters and Manned Lunar Survey Missions will decrease those uncertainties by an order of magnitude.

NAME	*LATITUDE (DEG)	*LONGITUDE (DEG)	**ALTITUDE (M)
Taruntius G	1.88	49.46	-1571
Messier B	-0.87	48.03	-1840
Toricelli C	-2.63	26.01	-352
E. Pickering	-2.86	7.01	1416
Rhaeticus A	1.73	5.20	3527
Gambart C	+3.32	-11.79	456
Turner	-1.38	-13.20	-54
Lansberg D	-3.01	-30.59	-940
Hortensius A	4.35	-30.69	-1669
Damoiseau E	-5.25	-58.30	-3219

*The lunar centered coordinate system is one whose axis are oriented to the moon's polar axis, equator, and mean libration point.
**Altitude is measured from a mean sphere of radius 1738.00 KM.

5.3 Communications Systems

The block diagram of Figure 5-1 presents the planned Apollo/Saturn ground communications network. The diagram is based on information from Appendix H, Apollo/SI-B PIRD (preliminary at time of writing). Some changes can therefore be anticipated.

Legend and notes:

TTY	Teletype, 60 words/minute (45.5 bits/second)
V	Voice, 0.3 to 3.0 kc/sec
V/D	Voice or data, 0.3 to 3.0 kc/sec

HSD	High speed data. 2400 bit/sec anticipated for KSC, ANT, BDA, CYI, CRO, HAW, GUM, GYM and TEX. 600 to 1200 bit/sec anticipated for ASC.
WBD	Wide band data; 40.8 kbit/sec
TV	TV-channel, 500 kc/sec bandwidth. Requirements for remoting TV from Goldstone to MSC will be satisfied via a commercial TV channel. Requirement for remoting TV from Canberra and Madrid is recommended to be fulfilled via NASCOM communication satellite system.
ALDS	Apollo Launch Data System
ANT	Antigua
ASC	Ascension Island
A Ship	Atlantic Ocean Ship
BDA	Bermuda
BER	Canberra Switching Station
BRA	Canberra
CAL	Point Arguello
CRO	Carnarvon
CSQ	Coastal Sentry Quebec
CTN	Canton Island
CYI	Grand Canary Island
EGL	Eglin
ETR	Eastern Test Range
GLD	Goldstone
GUM	Guam

GYM	Guyamas
HAW	Hawaii
HON	Honolulu Switching Station
I. O. Ship	Indian Ocean Ship
JPL	JPL Back-Up Stations
KSC	Cape Kennedy
LDN	London Switching Station
LIEF	Launch Information Exchange Facility
MAD	Madrid
PRE	Pretoria
P. Ship	Pacific Ocean Ship
RKV	Rose Knot Victor
TAN	Tananarive
TEX	Texas
WHS	White Sands

References

5-a. "Computing Notes on the First Gemini Mission GT-1," Data Operations Branch, Goddard Space Flight Center, Greenbelt, Maryland, April 27, 1964.

5-b. "Computing Notes on the Saturn Mission (SA-6)," Goddard Space Flight Center, Greenbelt, Maryland, June 18, 1964.

5-c. "Computing Notes on the Saturn Mission (SA-7)," Goddard Space Flight Center, Greenbelt, Maryland, September 23, 1964.

5-d. "Goddard Directory of Tracking Station Locations," Goddard Space Flight Center, Greenbelt, Maryland, July 1, 1964.*

5-e. "Bissett-Berman Apollo Note 265," Bissett-Berman Corporation, Santa Monica, California, October 7, 1964.

5-f. JPL Space Program's Summary NO 27-28, Vol III, pp. 2-6.

*This is the best information available at present and is now being used in all MSF, MSFC, and GSFC manned flight orbit determination programs. Official recognition of this document is pending.

Table 5-1

			Maximum Tracking Rate and Angular Coverage			Acquisition Times[1]			Operational Limits	Operational Accuracy, 1σ Values							
System	Band	Mount	Azimuth x Hour Angle	Elevation y Declination	Range	Angle Tracking	Range Rate	Range	Tracking	Azimuth x Hour Angle (m rad)		Elevation y Declination (m rad)		Range (feet)		Range Rate (feet/sec)	
										Noise	Bias	Noise	Bias	Noise	Bias	Noise	Bias
FPQ-6	C	AZ/EL	28°/sec	28°/sec	60,000 feet/sec	[2] 0	NA	[2] 0		0.15	0.3	0.15	0.3	20	40	NA	
TPQ-18			360°	-7° to 187°						0.2	0.4	0.2	0.4	30	60		
FPS-16	C	AZ/EL	40°/sec	30°/sec	Depending on Modification 24,000 or 96,000 feet/sec	[2] 0	NA	[2] 0		0.2	0.4	0.2	0.4	30	60	NA	
MPS-25			360°	-10° to 190°						1.0	2.0	1.0	2.0	60	120		
MPS 26	C	AZ/EL	56°/sec 360°	28°/sec -2° to 89.8°	30,000 feet/sec	[2] 0	NA	[2] 0		1.0	2.0	1.0	2.0	60	120	NA	
USBS 30' Antenna	S	X-Y	4°/sec ±88.0°	4°/sec ±80.0°		50 sec	50 sec	65 sec	Lunar distance with high gain antenna on spacecraft	0.8	1.6	0.8	1.6	60	120	[3] 0.10	[4] 0.07
USBS 85' Antenna	S	X-Y	3°/sec ±88.0°	3°/sec ±80.0°		50 sec	50 sec	65 sec	Lunar distance with high gain antenna on spacecraft	0.8	1.6	0.8	1.6	60	120	[3] 0.10	[4] 0.07
USBS 85' Antenna JPL BACK UP	S	HA/DEC				50 sec	50 sec	65 sec	Lunar distance with high gain antenna on spacecraft	0.8	1.6	0.8	1.6	60	120	[3] 0.10	[4] 0.07

1. Acquisition time required after spacecraft has reached 5° elevation
2. Acquisition normally occurs below 5° elevation
3. Integration period of 1 second assumed
4. 0.2 ft/sec for 3-way doppler

REMARKS:

Missing information to be added later.

The five tracking ships are omitted at present.

Table 5-2

Call Letters	Station	Site	Site Status	Operational Date	Geodetic Coordinates***			Geocentric Rectangular Coordinates		
					Latitude	Longitude	Height above ellipsoid (m)	U(m)	V(m)	W(m)
CNV	Cape Kennedy	*FPS-16	Operational	Now	28°28'54".36 ± 1".0	-80°34'35".45 ± 1".2	14 ± 40	918 608 ± 32	-5 534 781 ± 38	3 023 564 ± 34
		USBS 30' (Dual)	Planned	Feb 1967						
PAT	Patrick Air Force Base	FPQ-6	Operational	Now	28°13'35".59 ± 1".0	-80°35'57".45 ± 1".2	15 ± 40	918 602 ± 32	-5 548 399 ± 38	2 998 673 ± 34
GBI	Grand Bahama Island	FPS-16	Operational	Now	26°36'56".83 ± 1".0	-78°20'52".26 ± 1".2	14 ± 41	1 152 462 ± 32	-5 588 531 ± 39	2 840 241 ± 34
SSI	San Salvador Island	FPS-16	Operational	Now	24°07'08".37 ± 1".0	-74°30'14".68 ± 1".2	03 ± 42	1 556 158 ± 33	-5 612 875 ± 40	2 590 333 ± 34
GTI	Grand Turk Island	TPQ-18	Operational	Now	21°27'46".47 ± 1".0	-71°07'55".36 ± 1".2	25 ± 42	1 920 458 ± 33	-5 619 454 ± 40	2 319 191 ± 34
ANT	Antigua Island	FPQ-6	Operational	Now	17°08'37".67 ± 1".1	-61°47'33".66 ± 1".2	26 ± 42	2 881 624 ± 35	-5 372 559 ± 40	1 868 064 ± 34
		USBS 30'	Planned	Oct 1967						
BDA	Bermuda	*FPS-16	Operational	Now	32°20'51".96 ± 1".2	-64°39'13".12 ± 1".4	03 ± 43	2 308 919 ± 39	-4 874 348 ± 41	3 393 093 ± 39
		FPQ-6	Planned	July 1966						
		USBS 30'	Planned	Nov 1966						
CYI	Grand Canary Islands	1) MPS-26	Existing	May 1965	27°44'07".88 ± 4".6	-15°36'00".02 ± 5".1	29 ± 32	5 441 358 ± 77	-1 519 255 ± 138	2 950 625 ± 127
		USBS 30'	Planned							
ASC	Ascension Island	TPQ-18	Existing	Oct 1964	-07°58'22".78 ± 3".4	-14°24'06".10 ± 3".5	143 ± 32	6 118 552 ± 43	-1 571 171 ± 103	-878 847 ± 105
		FPS-16	Operational	Now						
		USBS 30' (Dual)	Planned	June 1967						
PRE	Pretoria, South Africa	MPS-25	Considered	Now	-25°56'43".99 ± 1".4	28°21'43".19 ± 1".5	1626 ± 43	5 051 390 ± 43	2 726 948 ± 43	-2 774 365 ± 43
CRO	Carnarvon, Australia	FPQ-6	Operational	Now	-24°53'50".48 ± 1".9	113°42'57".84 ± 2".2	64 ± 66	-2 328 319 ± 60	5 300 021 ± 64	-2 668 807 ± 60
		USBS 30' (Dual)	Planned	Nov 1966						
WOM	Woomera	FPS-16	Operational	Now	-30°49'10".96 ± 1".9	136°50'13".09 ± 2".2	173 ± 60	-3 998 927 ± 61	3 750 390 ± 61	-3 248 838 ± 60
GUA	Guam	2) USBS 30' (Dual)	Planned	May 1967	13°35'00".00 ± 6".4	144°55'30".00 ± 6".6	20 ± 32	-5 074 843 ± 125	3 563 353 ± 163	1 488 225 ± 193
HAW	Hawaii	**FPS-16	Operational	Now	22°07'30".96 ± 1".4	-159°40'03".43 ± 1".6	1142 ± 43	-5 543 977 ± 53	-2 054 341 ± 74	2 387 711 ± 73
		USBS 30' (Dual)	Planned	Feb 1967						
CAL	Pt. Arguello, California	FPS-16	Operational	Now	34°34'58".45 ± 1".0	-120°33'40".14 ± 1".2	646 ± 40	-2 673 158 ± 33	-4 527 065 ± 36	3 600 253 ± 34
GYM	Guaymas, Mexico	1) USBS 30'	Planned	April 1967	27°57'30".26 ± 1".0	-110°43'14".85 ± 1".2	18 ± 41	-1 994 778 ± 32	-5 273 239 ± 38	2 972 459 ± 33
WHS	White Sands, New Mexico	FPS-16	Operational	Now	32°21'29".60 ± 1".0	-106°22'10".43 ± 1".2	1232 ± 40	-1 520 192 ± 31	-5 175 317 ± 37	3 394 730 ± 33
TEX	Texas	1) USBS 30' (Dual)	Planned	Dec 1966	29°45'31".00 ± 1".0	-95°21'48".90 ± 1".1	50 ± 40	-518 010 ± 30	-5 517 391 ± 38	3 147 214 ± 32
	Houston									
EGL	Eglin Air Force Base	FPS-16	Operational	Now	30°25'18".36 ± 1".0	-86°47'53".21 ± 1".2	28 ± 40	307 465 ± 31	-5 496 185 ± 38	3 210 810 ± 33
MAD	Madrid, Spain	2) USBS 85' (Dual)	Planned	July 1967	40°25'00".00 ± 1".0	-03°40'00".00 ± 1".2	50 ± 43	4 852 944 ± 39	-310 991 ± 31	4 113 373 ± 37
		JPL 85' (Dual)	Planned (Standby)							
CNB	Canberra, Australia	2) USBS 85' (Dual)	Planned		-35°18'41".50 ± 1".9	149°08'09".00 ± 2".2	50 ± 66	-4 472 696 ± 63	2 673 056 ± 60	-3 666 173 ± 61
		JPL 85' (Dual)	Planned (Standby)	May 1967						
GST	Goldstone, California	USBS 85' (Dual)	Planned	Feb 1967						
		JPL 85' (Dual)	Existing (Standby)		35°23'22".70 ± 1".1	-116°50'55".60 ± 1".2	1031 ± 40	-2 351 393 ± 34	-4 645 137 ± 37	3 673 809 ± 35

*Mod 1
**Mod 2
***Fischer Ellipsoid 1960

Dual indicates a common antenna and two receivers and transmitters
1) Coordinates for existing Verlort
2) Preliminary coordinates

Table 5-3

Station	Site	Deflections of the Vertical (seconds)		Boresighting Data		Refractivity
		Meridian	Prime vertical	Azimuth	Elevation	
Cape Kennedy	FPS-16	-1 ± 1".0	1 ± 1".0	306°33'40" ± 30"	02°35'56" ± 30"	± 20"
	USBS 30'					
Patrick Air Force Base	FPQ-6	0 ± 1".0	1 ± 1".0	268°21'07" ± 30"	03°02'37" ± 30"	± 20"
Grand Bahama Island	FPS-16	-11 ± 1".0	5 ± 1".0	173°20'28" ± 30"	04°15'50" ± 30"	± 20"
San Salvador Island	FPS-16	13 ± 1".0	-5 ± 1".0	149°02'40" ± 30"	02°45'20" ± 30"	± 20"
Grand Turk Island	TPQ-18	11 ± 1".0	7 ± 1".0	169°43'25" ± 30"	02°48'43" ± 30"	± 20"
Antigua Island	FPQ-6	2 ± 2".0	10 ± 2".0	071°47'40" ± 30"	00°52'53" ± 30"	± 20"
	USBS 30'					
Bermuda	FPS-16	-14 ± 2".0	19 ± 2".0	282°44'41" ± 30"	01°59'23" ± 30"	± 20"
	FPQ-6					
	USBS 30'					
Grand Canary Islands	MPS-26	-31 ± 5".0	4 ± 6".0			
	USBS 30'					
Ascension Island	TPQ-18	-2 ± 4".0	-4 ± 4".0			
	FPS-16					
	USBS 30' (Dual)					
Pretoria, South Africa	MPS-25			035°46'14" ± 30"	01°36'16" ± 30"	± 20"
Carnarvon, Australia	FPQ-6			138°28'00" ± 30"	02°02'00" ± 30"	± 20"
	USBS 30' (Dual)					
Woomera	FPS-16	1 ± 1".0	-1 ± 1".0	72°26'58" ± 30"	2°38'25" ± 30"	± 20"
Guam	USBS 30' (Dual)					
Hawaii	FPS-16	12 ± 2".0	-13 ± 2".0	202°43'42" ± 30"	00°21'27" ±30"	± 20"
	USBS 30' (Dual)					
Pt. Arguello, California	FPS-16	-5 ± 1".0	-8 ± 1".0	287°36'56" ± 30"	351°29'12" ± 30"	± 20"
Guaymas, Mexico	USBS 30'	-2 ± 1".0	-10 ± 1".0			
White Sands, New Mexico	FPS-16	0 ± 3".0	-1 ± 3".0	185°32'16" ±30" *(-00°01'09")	02°27'23" ±30"	± 20"
Texas	USBS 30' (Dual)					
Eglin Air Force Base	FPS-16	0 ± 1".0	-1 ± 1".0	355°31'39" ± 30"	01°27'37" ± 30"	± 20"
Madrid, Spain	USBS 85' (Dual)					
	JPL 85' (Dual)					
Canberra, Australia	USBS 85' (Dual)					
	JPL 85' (Dual)					
Goldstone, California	USBS 85' (Dual)					
	JPL 85' (Dual)					

*Additional correction for true north is required at this station.

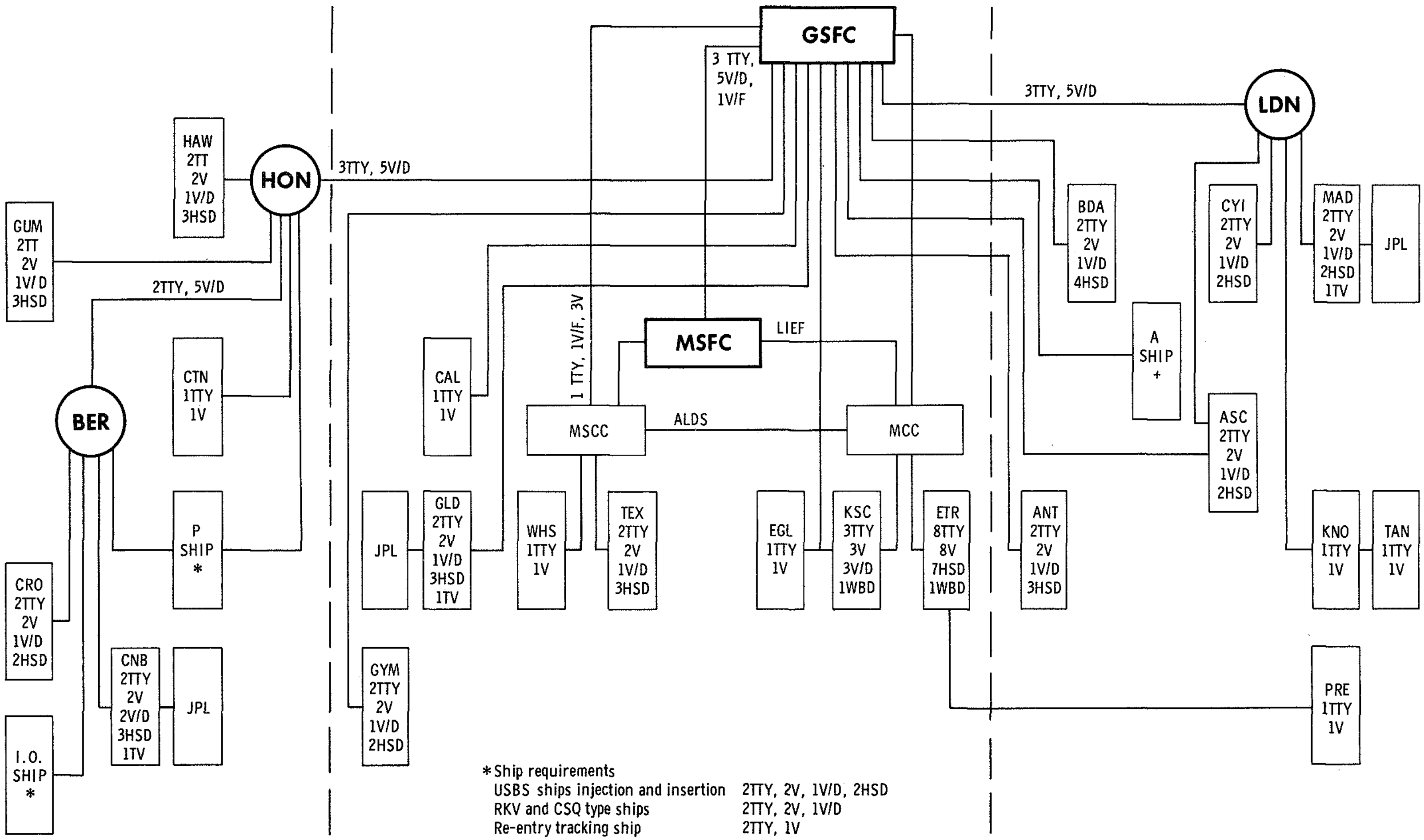

Figure 5-1—Apollo/Saturn-V Ground Communications Network

6.0 DATA PROCESSING PROCEDURES

This chapter describes those aspects of ground based orbit determination procedures and methods which influence navigation capability studies. The information is included because the sophistication of the programs and techniques used and the procedures for processing influence the accuracy and speed with which the orbit may be determined.

At present, no variables other than the position and velocity will be determined in the orbit determination process.